TECHNICAL REPORT NO. LWL-CR-02M69

HELICOPTER PAYLOAD CAPABILITY INDICATOR

Final Report
Contract No. DAAD05-68-C-0366

By

E. Kisielowski
E. Fraundorf

Scientific Systems Division
Dynasciences Corporation
Blue Bell, Pennsylvania

March 1971

APPROVED FOR PUBLIC RELEASE; DISTRIBUTION UNLIMITED.

U. S. ARMY LAND WARFARE LABORATORY

Aberdeen Proving Ground, Maryland 21005

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ABSTRACT

A feasibility study was made of a helicopter payload meter concept. A simple, manually operated device was developed and tested, which gives an indication of payload capability in terms of gas generator speed for the prevailing atmospheric conditions where vertical take-offs and landings are required from a confined area. Tests were conducted by the U.S. Army Aviation Test Board, and the device was found to have "military potential."

FOREWORD

The work was sponsored by the U. S. Army Land Warfare Laboratory, Aberdeen Proving Ground, Maryland and was performed by the Scientific Systems Division of Dynasciences Corporation, Blue Bell, Pennsylvania, under Contract No. DAAD05-68-C-0366 (Tasks 07-MA-68 and 02-M-69) during the period from May 1968 to November 1969.

The Army technical representatives were Mr. C. Wilson, LTC. D. Haller, and Mr. E. Vanderlip. The contributions of the Army technical personnel to this work are gratefully acknowledged.

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SYMBOLS

A	rotor disc area (total disc area of two rotors for tandem rotor helicopter), ft^2
C_P	rotor power coefficient, $P/\rho A(\Omega R)^3 = P/(PF)$
C_T	rotor thrust coefficient, $T/\rho A(\Omega R)^2 = T/(TF)$
C_W	weight coefficient $W/\rho A(\Omega R)^2 = W/(TF)$
F,f G,g H	designate functions
h_p	pressure altitude, ft
M_T	advancing blade tip Mach number, $(V+\Omega R)/Vc$
N	rotor (or engine) rpm
P	rotor power, ft-lb/sec
p	atmospheric pressure at any altitude, lb/ft^2
p_o	sea level standard atmospheric pressure, 2117 lb/ft^2
PF	power factor, $T_F(\Omega R)$
PL	payload, $(W_G)_{MAX} - W_o$, lb
Q	rotor (or engine) torque, ft-lb
R	rotor radius, ft
SHP	rotor shaft horsepower
$(SHP)_{TOTAL}$	total shaft horsepower
T	rotor thrust, lb
TF	thrust factor, $\rho A(\Omega R)^2$
t	temperature ^oF

t_o	sea level standard atmospheric temperature, 59°F
V	aircraft forward speed, ft/sec
V_T	rotor tip speed, ΩR, ft/sec
V_c	ambient speed of sound, ft/sec
$(W_G)_{MAX}$	maximum gross weight, lb
W_o	initial weight before loading
δ	relative atmospheric pressure, p/p_o
θ	relative atmospheric temperature, t/t_o
μ	rotor tip-speed ratio, V/V_T
ρ	air density, slugs/ft^3
ρ_o	sea level standard air density, 0.002378 slugs/ft^3
σ	relative air density, ρ/ρ_o
Ω	rotor rotational speed, rad/sec

I. INTRODUCTION

Helicopter operations in remote areas away from the instrumentation of the normal airfield are often constrained by the lack of knowledge of the helicopter payload capability. This payload capability is defined as a difference between maximum helicopter weight achievable at maximum power available and the initial weight prior to loading. The payload carrying capability of a helicopter is primarily dependent on rotor performance and engine power available, which are directly related to local air density. In turn, the local air density is a function of the environmental atmospheric conditions such as ambient air temperature, pressure altitude and air humidity.

The basic problem in devising an accurate payload indicator (within ± 200 lb) is associated with difficulties in obtaining accurate measurements of ambient air temperature and pressure altitude which determine the required air density. An additional parameter which affects helicopter payload capability is the aircraft initial weight before loading. Although consideration is given in this study to the determination of aircraft initial weight, it is herein assumed that this parameter is a known input.

The prime objective of this work assignment (under contract No. DAAD05-68-C-0366) was to evaluate the effects of these parameters on helicopter lifting capability and to

perform a comprehensive feasibility study of the helicopter payload meter concept. As a first step of this study a simple, manually generated GO-NO-GO payload indicator for the UH-1H helicopter was designed, fabricated and tested. This device provided the pilot with valuable information on the aircraft's ability to safely complete a vertical take-off and landing maneuver as affected by its current loading and the existing atmospheric conditions.

The following sections present the results of the feasibility study based on the performance data of the CH-47A helicopter, and a description of the GO-NO-GO payload indicator designed for the UH-1H helicopter, together with the operating instructions and supporting data.

II. FEASIBILITY STUDY OF HELICOPTER PAYLOAD CAPABILITY METER

Presented in this section are the results of a feasibility study of a helicopter payload meter concept. The numerical results obtained in this study are specifically applicable to the CH-47A (Chinook) helicopter. However, the design approaches and methods developed herein are equally well applicable to other U. S. Army helicopters.

The payload meter concept considered in this study combines meteorological data, engine and aircraft performance characteristics and aircraft initial weight before loading into a single visual display of pounds payload capability.

The meteorological data consists of air ambient temperature and pressure altitude measurements which can be directly obtained from instrumentation on board the aircraft; i.e., temperature indicator and altimeter, respectively.

The engine data comprises engine maximum power available (military rating) as a function of ambient temperature and pressure altitude. The engine power available is limited by the ram power or gearbox transmission limits whichever occurs sooner. The aircraft performance characteristics required to determine aircraft payload capabilities are the basic power required to hover versus weight relationships. This information is most suitably obtained from the test data presented in the form of a nondimensional C_T vs. C_p curve. An additional

parameter required for determining aircraft payload capabilities is the aircraft initial weight (W_o) before loading. Although the aircraft initial weight can be established by a simple hovering test (as will be described later in the text), it will be assumed for the purpose of this study that this parameter is a known or a given input.

The principle of the payload capability meter can be best illustrated by utilizing the actual helicopter performance, engine and meteorological data of an existing helicopter. As mentioned above, the sample helicopter considered in this study is the Boeing CH-47A (Chinook) equipped with two T-55-L-7 Lycoming engines.

Presented below are the numerical procedures for determining the helicopter payload capability given the initial weight before loading. Also, a method is presented for obtaining the initial aircraft weight from a simple hovering test. Implementation of these procedures into an analog display of the helicopter payload capability and the overall accuracy of the resulting payload meter, are also discussed.

A. DETERMINATION OF HELICOPTER PAYLOAD

1. Obtain the following input parameters:
    (a) Ambient air temperature $t(^{o}F)$ - from onboard temperature indicator
    (b) Pressure altitude h_p (ft) from altimeter

(c) Engine power available $\left[\, SHP = F(t,\ h_p)\right]$ from engine specification (Military rated power)

(d) Initial helicopter weight (W_o) before loading

(e) Helicopter hover performance $\left[C_T = f(C_P)\right]$

(f) Helicopter design parameters and operating conditions, i.e., ρ_o , πR^2, ΩR

2. Knowing ambient air temperature and pressure altitude (from step (1)), enter charts (Figure 1) and obtain air density ratio σ. Alternatively, compute air density ratio σ using the following equation:

$$\sigma = \left[\frac{288.16}{5/9\ (t^{o}F-32) + 273.16}\right]\left[1 - \frac{0.001981\ h_p}{288.16}\right]^{5.256}$$

3. Knowing $t(^{o}F)$ and $h_p(ft)$ (from step (1)), enter Figure 2 and obtain maximum total engine power available (military rating). Use actual total shaft horsepower, available $(SHP)_{TOTAL}$ or transmission limit power, whichever is lower.

4. Using helicopter design parameters and operating conditions (from step (1)), compute

$$TF = \rho_o \pi R^2 (\Omega R)^2 \quad \text{and} \quad PF = TF\ (\Omega R)$$

5. Compute power coefficient using σ from step (2), $(SHP)_{TOTAL}$ from step (3) and PF from step (4) as follows:

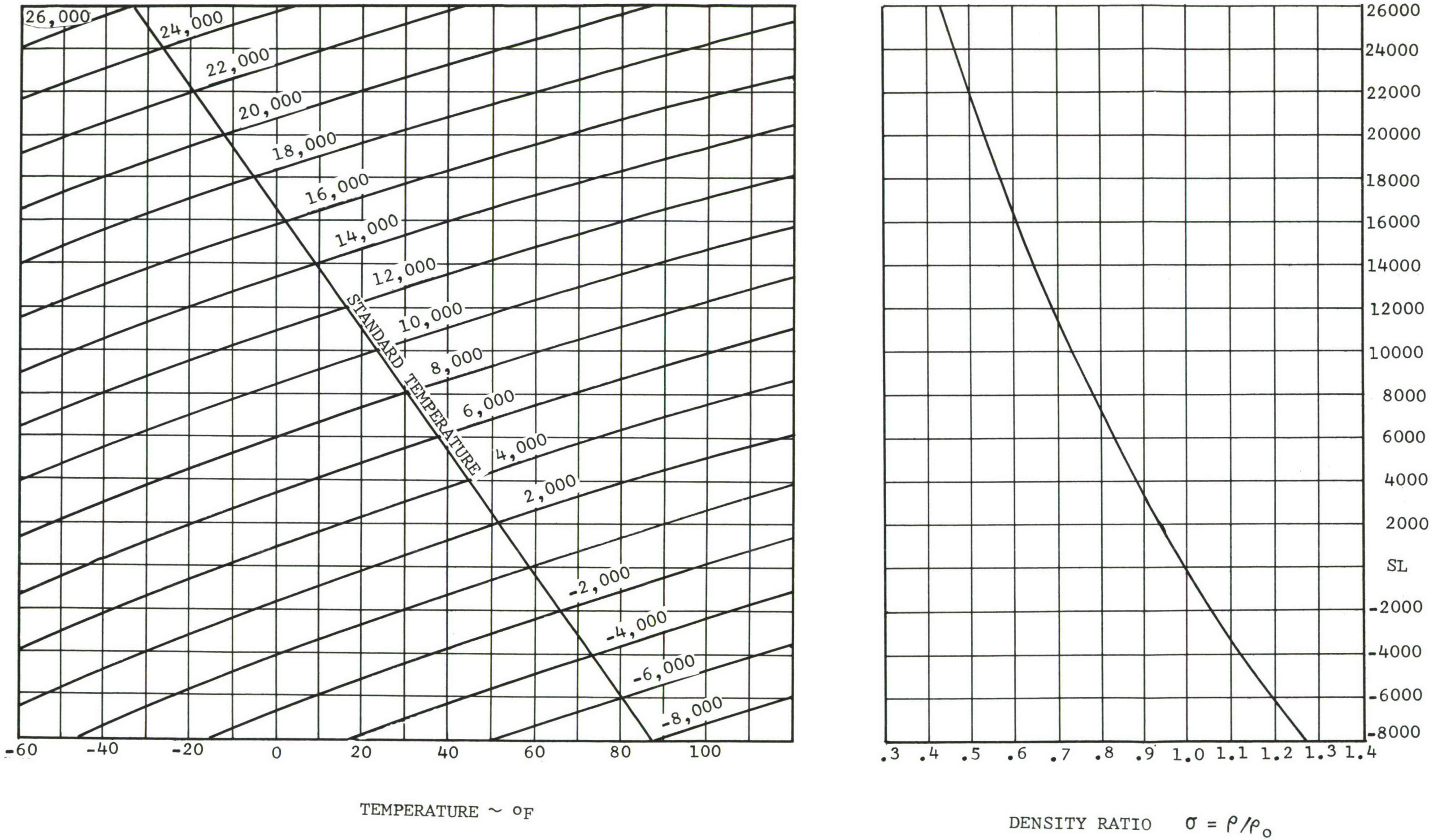

Figure 1. Density Altitude Chart.

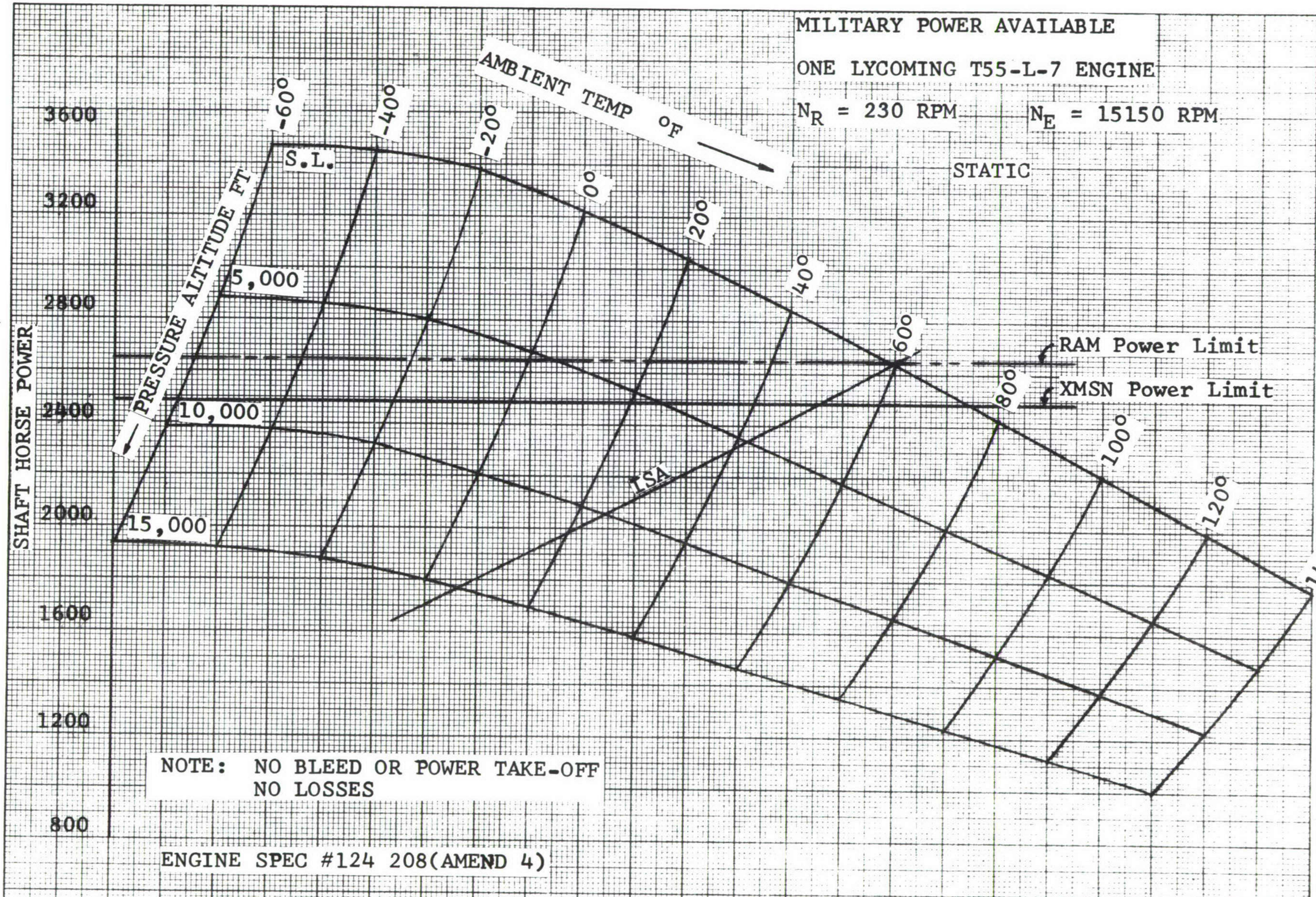

Figure 2. Engine Power Available (Military Rating).

$$C_P = \frac{(SHP)_{TOTAL}}{\sigma} \frac{550}{(PF)}$$

6. Using C_p from step (5), enter Figure 3 and obtain the corresponding C_W.
7. Using C_W from step (6), σ from step (2) and TF from step (4), compute helicopter maximum weight capability thus,

$$(W_G)_{max} = C_W \, \sigma \, (TF)$$

8. Compute payload using W_o from step (1) and $(W_G)_{max}$ from step (7) thus,

$$(PL) = (W_G)_{max} - W_o$$

9. Display on a dial or a digital readout the value of PL from step (8).

The above procedure was utilized to determine maximum takeoff gross weight for the CH-47A helicopter. The numerical results thus obtained are presented in a "carpet" plot, Figure 4.

B. DETERMINATION OF AIRCRAFT INITIAL WEIGHT (W_o)

1. Obtain the following input parameters:
   (a) Helicopter hovering performance $\left[C_W = f(C_p)\right]$
   (b) Helicopter design and operating conditions of

$$\rho_o, \pi R^2, (\Omega R)$$

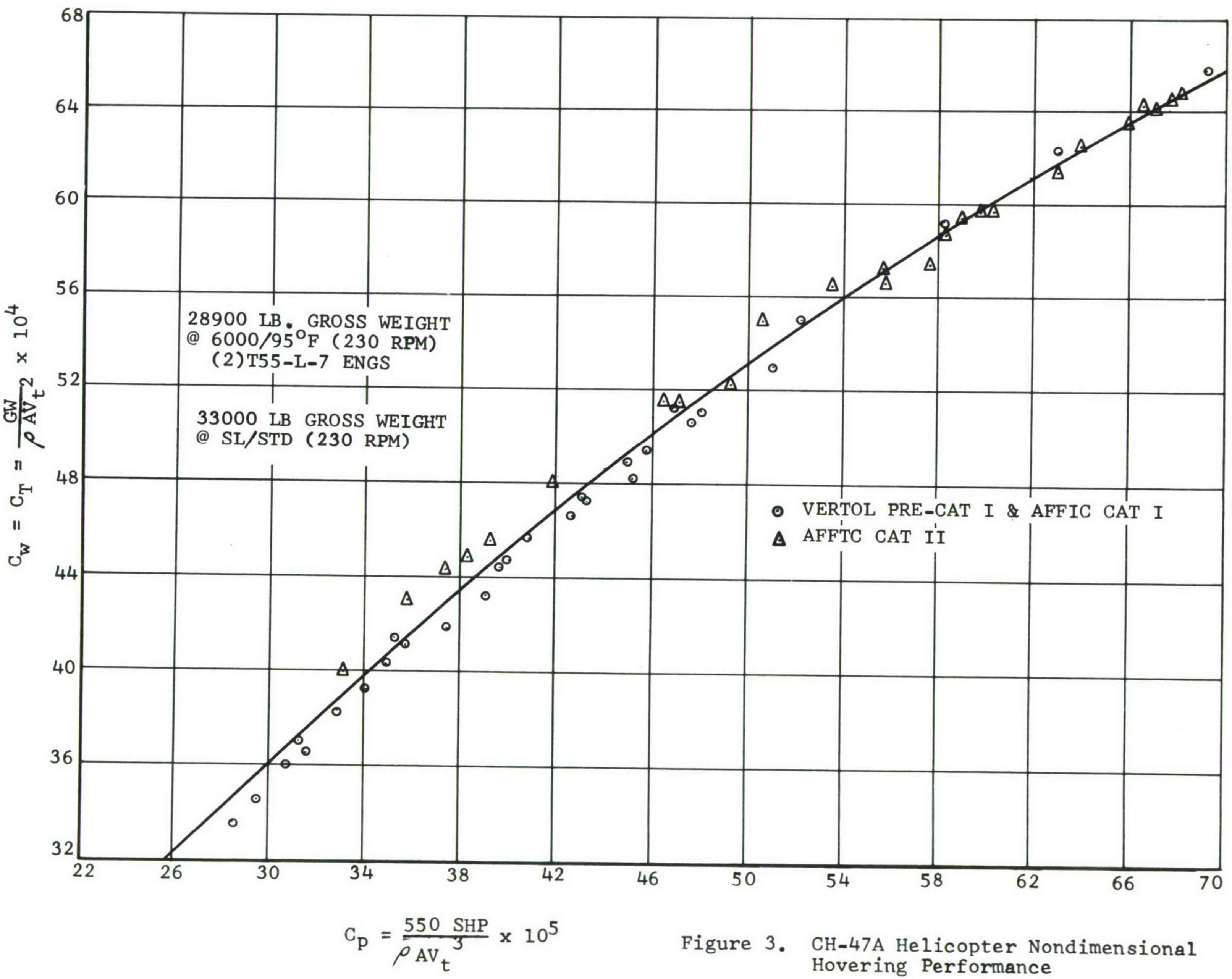

Figure 3. CH-47A Helicopter Nondimensional Hovering Performance

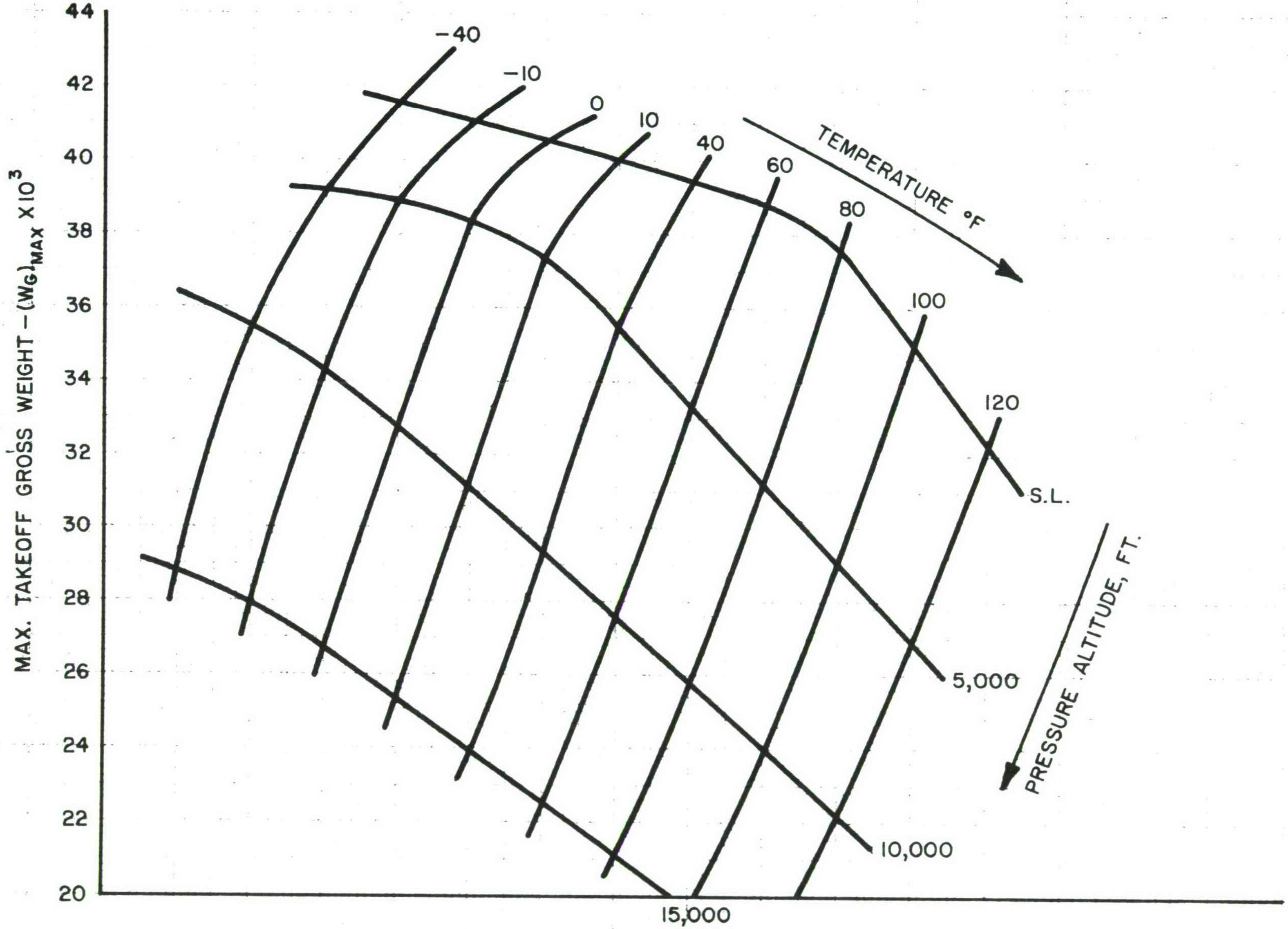

Figure 4. CH-47A Helicopter Maximum Takeoff Weight Capabilities.

2. Perform a simple hovering test at any altitude out of ground effect and record:

   (a) Ambient temperature $t(^{o}F)$

   (b) Pressure altitude h_p (ft)

   (c) Engine torque Q(ft-lb)

   (d) Shaft rpm - N

   Note: This test should be performed at a low enough altitude (out of ground effect) to permit quick aircraft landing without excessive fuel burnoff.

3. Knowing ambient temperature and pressure altitude from step (2), enter Figure 1 and obtain air density ratio σ. Alternatively compute air density ratio using the equation given in subsection (A), step (2).

4. Knowing engine torque Q and the corresponding rpm N, from step (2) above, compute total shaft horsepower required to hover, thus:

$$(SHP)_{TOTAL} = \frac{\pi \; N \; Q}{16,500}$$

5. Using helicopter design parameters and operating conditions, step (1) above, compute

$$TF = \rho_o \pi R^2 (\Omega R)^2$$

and

$$PF = TF \; (\Omega R)$$

6. Using σ from step (3), $(SHP)_{TOTAL}$ from step (4) and PF from step (5), compute the power coefficient required to hover, thus:

$$C_p = \frac{(SHP)_{TOTAL}\ 550}{\sigma \quad (PF)}$$

7. Using the power coefficient from step (6), enter Figure 3 and obtain the aircraft hovering weight coefficient (C_W).
8. Compute aircraft initial weight (W_o) using C_W from step (7) and TF from step (5), thus,

$$W_o = C_W\, \sigma\ (TF)$$

9. Store the value of W_o obtained in step (8) and enter it as an input to the payload indicator discussed in Section A above. Alternatively, display W_o, preferably on the same dial or digital readout as the payload.

C. ELECTRICAL DISPLAY OF HELICOPTER PAYLOAD

There are several approaches which could be utilized to implement the procedures derived in the preceding sections. These could be divided into the two broad classifications of analog and digital computations. Although it is believed that a digital approach would be more accurate and more flexible (readily applicable to different helicopters), it is expected to be more expensive as compared to analog. If, in addition to determining payload (knowing W_o), the initial aircraft weight

(W_o) is to be determined as described in Section B, then the digital approach may be more economical. A more detailed study of the tradeoff between the two approaches will be performed in the hardware part of the program whereby an optimum system will be selected. For the purpose of this feasibility study of a payload capability meter knowing aircraft initial weight, an analog approach is herein selected as described below.

Figure 5 shows a block diagram representation of this analog computer. In this computer both temperature and altitude information would be applied to function generators in order to develop the terms of the equation for σ. The output of one generator is proportional to the term

$$F(g) = \frac{288.16}{5/9(t-32) + 273.16}$$

and the other to the term

$$G(g) = \left[1 - \frac{.001981\ h_p}{228.16}\right]^{5.256}$$

These two terms are then multiplied together to give σ.

The next step is to determine the power available through the use of Figure 2. A study of this figure revealed that the shaft horsepower available per engine could be closely approximated analytically as SHP = $f(h_p)\ f(t)$. It appears that such a representation would have an accuracy of the order of ½ percent. This is the process indicated in the block diagram (Figure 5). Both temperature (t) and pressure altitude (h_p) signals

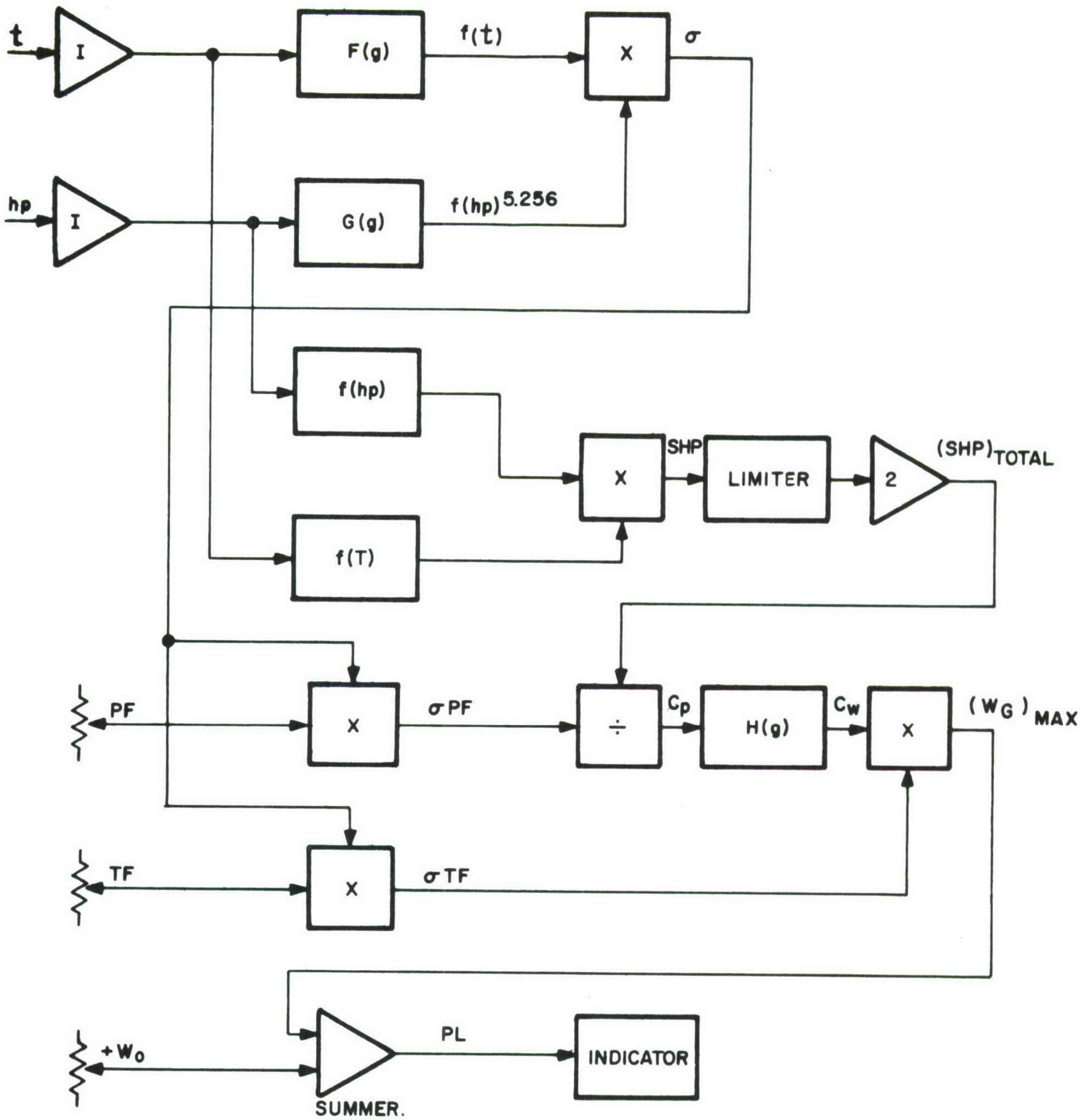

Figure 5. Analog Computer Schematic.

are applied to function generators and the outputs multiplied together to give a signal proportional to SHP per engine. The resulting output is then applied to a limiter circuit and multiplied by a factor of 2 to generate $(SHP)_{TOTAL}$.

Provisions are made to preset PF, TF, and W_o into the computer since these are constants for any given helicopter. Some thought was given to calculating PF and TF for an arbitrary helicopter but the complexity and extra hardware involved does not appear to warrant the increased size and cost of the system.

As indicated in the diagram, both PF, and TF are multiplied by σ. The factor σ(PF) is divided into the SHP_{TOTAL} term to give a signal proportional to C_p. This signal is then applied to another function generator H(g) to generate C_W. Multiplication of C_W by σ(TF) yields a signal proportional to $(W_G)_{max}$ which is compared to W_o in order to determine the allowable payload. This signal is then applied to the pilot's indicator to provide a visual display of the aircraft payload capability.

It is estimated that the electronics for a system such as the one described could be built into a volume of approximately 200 cu. in. This volume does not include the indicator. Using analog computation techniques as indicated, the computed payload capability would probably have an error between 1 and 2%. This analog error is taken into account in computing final

instrument accuracy as discussed in Section D below.

D. ACCURACY OF THE PAYLOAD CAPABILITY METER

The accuracy of the helicopter payload capability meter is primarily dependent upon the accuracy of the available instrumentation to provide basic inputs of ambient air temperature and pressure altitude, the accuracy of initial weight input, and the accuracy of electronic implementation of the system.

In order to provide an indication of the accuracy of the payload capability meter described in the previous section, an error analysis applicable to the CH-47A (Chinook) helicopter is herein performed.

1. The CH-47 Instrumentation Accuracy

The instrumentation on the CH-47A test helicopter used to obtain the hovering performance test data presented in Figure 3 was as follows:

(a) Altimeter

The CH-47A altimeter was an aneroid barometer Model No. Kollsman 671BK010B. This instrument meets the requirements of the MIL-A-6863D (amendment 2) specifications and yields the pressure altitude measurement within the following accuracy:

Pressure Altitude, ft	0	5000	10,000	15,000	20,000
Maximum Error, ft	±64	±75	±90	±115	±152

The production CH-47 helicopters are equipped (or are being equipped) with more accurate altimeters - Rosemount Transducer type with the maximum voltage output of 5 volts, yielding a maximum error of ±50 ft up to 10,000 ft of pressure altitude.

(b) Temperature Indicator

The temperature indicator was a Lewis Model 5525 resistance bridge with a maximum voltage output of 5 volts. The temperature dial was a Lewis Model 102330 with a maximum range of ±50°C. This instrument is capable of providing temperature measurements within ±2°C. The temperature indicator installed on production aircraft are expected to provide an increased accuracy within ±1°F.

(c) Rotor Shaft Horsepower

Total rotor shaft horsepower (Figure 3) was obtained from rotor shaft torque (within ±1%) and rotor rpm measurements (within ±0.07%). The rotor rpm was obtained by a tachometer Model GE U7/A.

2. Helicopter Performance Degradation

The CH-47A performance degradation is considered negligible and well within the accuracy of the instrumentation.

3. Engine Power Available

Engine power available (Figure 2) includes a 3 to 5 percent power loss due to engine degradation. Therefore,

Figure 2 represents a conservative military rated power for the Lycoming T-55-L-7 engine used on the CH-47 helicopters. The engine power loss is restorable by overhauling the engines after 1000 to 1800 hours of continuous operation. It should be noted that a considerable power loss can result from excessive sand and dust ingestion by the engines. This factor, however, cannot be determined under this study.

According to the engine manufacturer, the engine output shaft torque can be recorded within ±4%.

4. Air Humidity

An investigation was performed to determine the effect of air humidity on the air density computation. Although air humidity may affect air density to the extent of ±0.5%, both engine and aircraft manufacturers consider this effect to be negligible and well within the accuracy of the available instrumentation.

5. Payload Meter Maximum Error

The instrumentation errors discussed above are reflected in the scatter of the experimental data points (Figure 3) defining helicopter hovering performance. Examining the maximum scatter relative to the RMS curve (solid curve) it can be noted that the CH-47A hovering performance is measured within ±3% accuracy, which corresponds to approximately ±600 to ±900 lb. error based on the aircraft normal gross weight.

In order to determine the accuracy of the payload meter,

it is assumed the CH-47A RMS performance curve (solid line of Figure 3) is exact. Also curve reading and computational errors using the RMS curve are neglected. The payload meter accuracy, as can be noted from Figure 4, would then be a function of temperature and pressure inputs and electronic circuitry required for visual display of the payload.

Examining Figure 4, it can be noted that the maximum rate of change of $(W_G)_{max}$ with temperature t^oF at constant pressure altitude occurs at low pressure altitude (S.L.) and high temperature (100^oF to 120^oF). This rate of change is given by:

$$\left[\frac{\partial (W_G)_{max}}{\partial t}\right]_{h_p} = -137 \text{ lb}/^oF$$

Similarly, the maximum rate of change of $(W_G)_{max}$ with pressure altitude at constant temperature (which occurs at high pressure altitude and low temperature) is given by:

$$\left[\frac{\partial (W_G)_{max}}{\partial h_p}\right]_t = -1.3 \text{ lb/ft}$$

Now using production CH-47A instrumentation errors of $\pm 1^oF$ in temperature and ± 50 ft in pressure altitude (as discussed previously) and assuming 2.0% analog computer error, the payload meter accuracy for 10,000 lb of payload can be computed as follows:

$$\text{Maximum total error} = \pm\sqrt{(137 \times 1)^2 + (1.3 \times 50)^2 + (0.02 \times 10{,}000)^2}$$

$$= \pm\ 250.9 \text{ lb}$$

If the digital instead of analog approach is utilized in determining aircraft payload, the maximum total error of the payload indicator would be:

$$\text{Maximum total error} = \pm\sqrt{(137 \times 1)^2 + (1.3 \times 50)^2} = \pm\ 151.6 \text{ lb}$$

III DESIGN AND FABRICATION OF THE GO-NO-GO PAYLOAD INDICATOR

Presented in this section is the design description of the GO-NO-GO payload indicator together with the pertinent design and operating instructions for the instrument.

A. DESCRIPTION

The GO-NO-GO payload indicator shown in Figure 6 is a simple manually adjustable instrument requiring no power for its operation. The device determines the operational limits of the gas producer speed (N_1) of the helicopter turbine for the predetermined flight modes at given temperature settings. An internal adjustment is provided which allows "zeroing" of the device, i.e. setting standard day, placard N_1.

The device is designed for the UH-1H helicopter and is utilized to define the maximum usable gas producer speed N_1 for the existing gross weight to allow a sufficient power margin for the safe execution of:

(a) Vertical take-off and climb from a 2.0 ft. skid height hover to out-of-ground-effect hover.

(b) Normal take-off from a 2.0 ft. skid height hover.

(c) Out-of-ground-effect hover and vertical descent from a forward flight velocity of 55-65 knots.

(d) A 2.0 ft. skid height hover from a forward velocity of 55-65 knots.

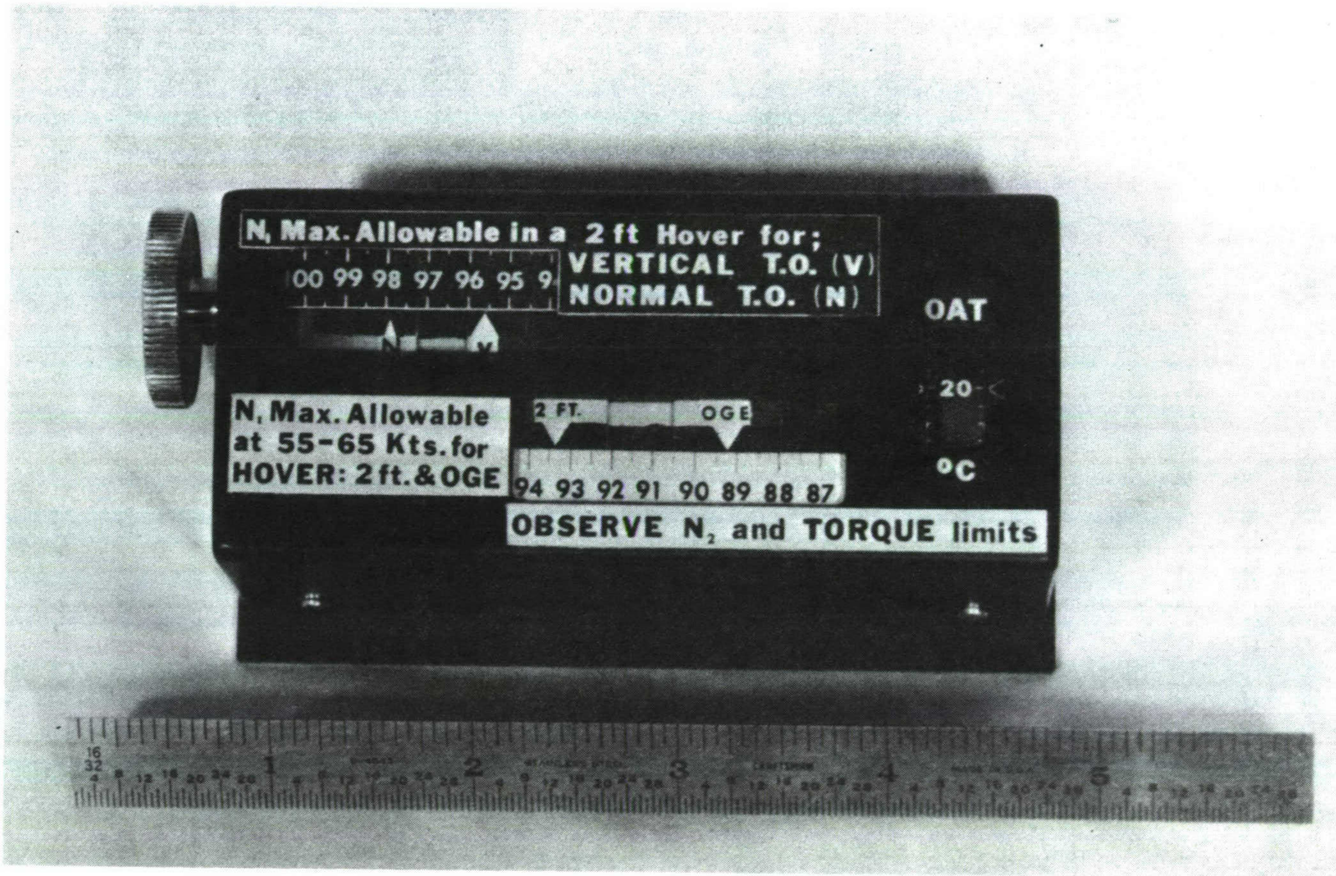

Figure 6. GO-NO-GO Indicator

It should be noted that since the indicator is essentially based on specific flight modes, conditions will exist where the indicated maximum usable N_1 is exceeded, i.e. a no-go condition. However, safe take-offs and landings may still be conducted by using recommended standard flight techniques, e.g. utilizing prevailing winds or performing a running take-off.

The payload meter is a mechanization of the UH-1H helicopter performance and engine power data combined into a visual display presenting N_1 as a function of ambient air temperature. As can be seen from Figure 6 the device has two scales; the upper scale given values of N_1 applicable to take-off while the lower scale is used for landing. On the flight article the scales and indicators are color-coded for ease of interpretation, i.e. green for go, yellow for caution and red for no-go.

The take-off scale is black with silver lettering. Its left hand indicator gives the "red line" or standard day placard N_1, appearing as a vertical red boundary at the extreme left of the scale. The maximum usable (allowable) N_1 at a 2.0 ft. skid height hover which will allow for a normal take-off, ("red line" less 2%) is represented by the small marker labeled N. The right hand indicator, marked V, shows the maximum usable (allowable) N_1 at a 2.0 ft. skid height hover and provides sufficient reserve power for a safe vertical climb to out-of-ground-effect hover. The scale is color-coded green

to the right of V; yellow between V and N and red to the left of N.

The lower scale is silver with black lettering. The left hand indicator, marked 2.0 ft., gives the maximum usable (allowable) N_1 in the velocity range of 55-65 knots which will provide reserve power for a 2.0 ft. skid height hover. The right-hand indicator, marked OGE, gives the maximum usable N_1 for this velocity range which allows sufficient power for transition to an OGE hover. This scale is color-coded green to the right of OGE, yellow between 2.0 ft. and OGE and red to the left of the 2.0 ft. marker.

The temperature as applied to this device is in actuality the turbine inlet temperature. However, for all practical applications this is equivalent to the outside air temperature (OAT) measured by the cockpit thermometer.

The temperature setting for forward velocity should be accomplished at an altitude range of 200-500 feet to minimize the temperature differential between the OGE and IGE flight modes.

B. DESIGN DATA

The design data used for this program was obtained from References 1 through 6. The engine power and rotor performance data extracted from these references was utilized to relate the gas producer turbine speed, N_1, and temperature for several ambient conditions and flight modes.

Three flight modes were investigated for the UH-1H helicopter and the power requirement of each defined as a function of temperature. These were OGE hover, IGE hover at a 2.0 ft. skid height, and a forward velocity between 55 and 65 knots. Having obtained these power requirements it was possible to determine the incremental power between the required conditions, i.e. (OGE-IGE), (OGE-55/65 knots) and (IGE-55/65 knots) which represent the reserve power between the specified flight modes. These are coupled with the engine temperature bias rating curve to obtain reserve power relative to 100% N_1.

The power data used in the design of the payload meter indicator was obtained from the T-53-L-13 turbine specification of Reference 1. Specifically, this turbine altitude performance data (N_1 versus SHP for various altitudes) was normalized to standard sea level conditions. Normalization to standard sea level conditions consisted of correcting the SHP for pressure and temperature for the specific altitude: i.e. $SHP/\delta\sqrt{\theta}$ where δ is the pressure ratio $\frac{p}{p_o}$ and θ is the temperature ratio $\frac{t}{t_o}$. The calculation of this data is presented in Table I and the final results are plotted in Figure 7.

Since this curve was obtained from the basic engine specification data there is no correction for turbine air bleed, accessories, or installation losses. However, these corrections are included in the engine operating limits curve shown in Figure 8.

Utilizing the results of Figures (7) and (8), the following

Referred SHP vs N_1 i.e. $SHP/\delta\sqrt{\theta}$ vs $N_1/\sqrt{\theta}$

N_1 and SHP data obtained from engine spec manual for T.O. Power - Estimated Data

SEA LEVEL; δ =1.0, T=518.4^{o}R, $\sqrt{\theta}$=1, $1/\delta\sqrt{\theta}$=1						
N_1	25400	24550	23750	22875	22050	21000
SHP	1400	1200	1000	800	600	400
$SHP/\delta\sqrt{\theta}$	1400	1200	1000	800	600	400
$N_1/\sqrt{\theta}$	25400	24550	23750	22875	22050	21000
5000 ft; δ =.832, T=500.57, $\sqrt{\theta}$=.9828, $1/\delta\sqrt{\theta}$=1.22314						
N_1	25400	24550	24050	23150	22220	21150
SHP	1255	1100	1000	800	600	400
$SHP/\delta\sqrt{\theta}$	1535	1345.5	1223	978.5	733.9	499
$N_1/\sqrt{\theta}$	25844	24929	24471	23555	22609	21500
10000 ft; δ =.6876, T=482.74, $\sqrt{\theta}$=.9650, $1/\delta\sqrt{\theta}$=1.50711						
N_1	25400	24700	23600	22540	21300	
SHP	1120	1000	800	600	400	
$SHP/\delta\sqrt{\theta}$	1688	1507	1250.7	904	603	
$N_1/\sqrt{\theta}$	26321	25595	24456	23357	22073	
15000 ft; δ =.5642, T=464.9, $\sqrt{\theta}$=.9470, $1/\delta\sqrt{\theta}$=1.87160						
N_1	25450	24100	22800	21450		
SHP	975	800	600	400		
$SHP/\delta\sqrt{\theta}$	1825	1497	1123	748		
$N_1/\sqrt{\theta}$	26874	25449	24076	22650		

Table 1. Referred SHP vs N_1

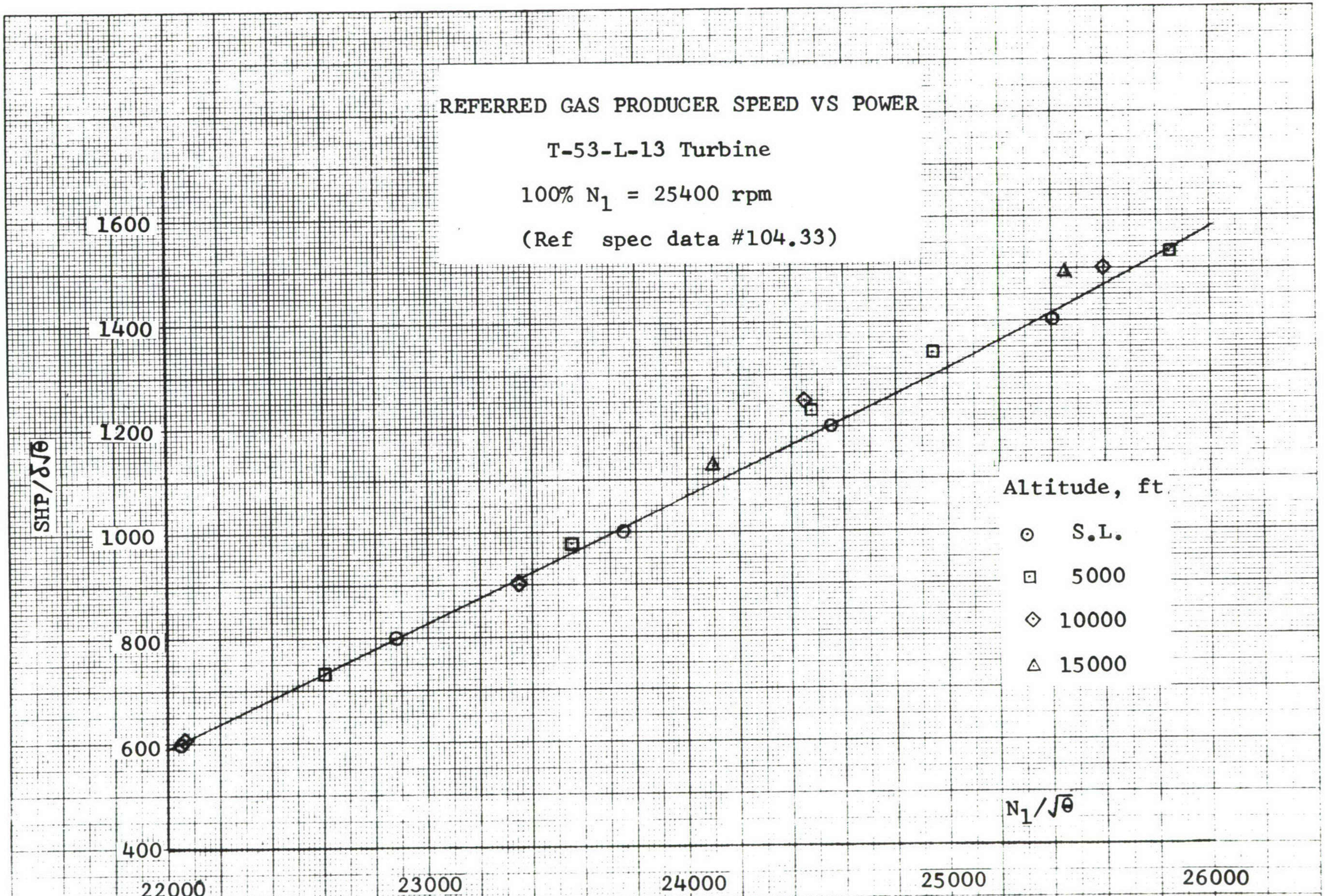

Figure 7. Referred SHP vs N_1

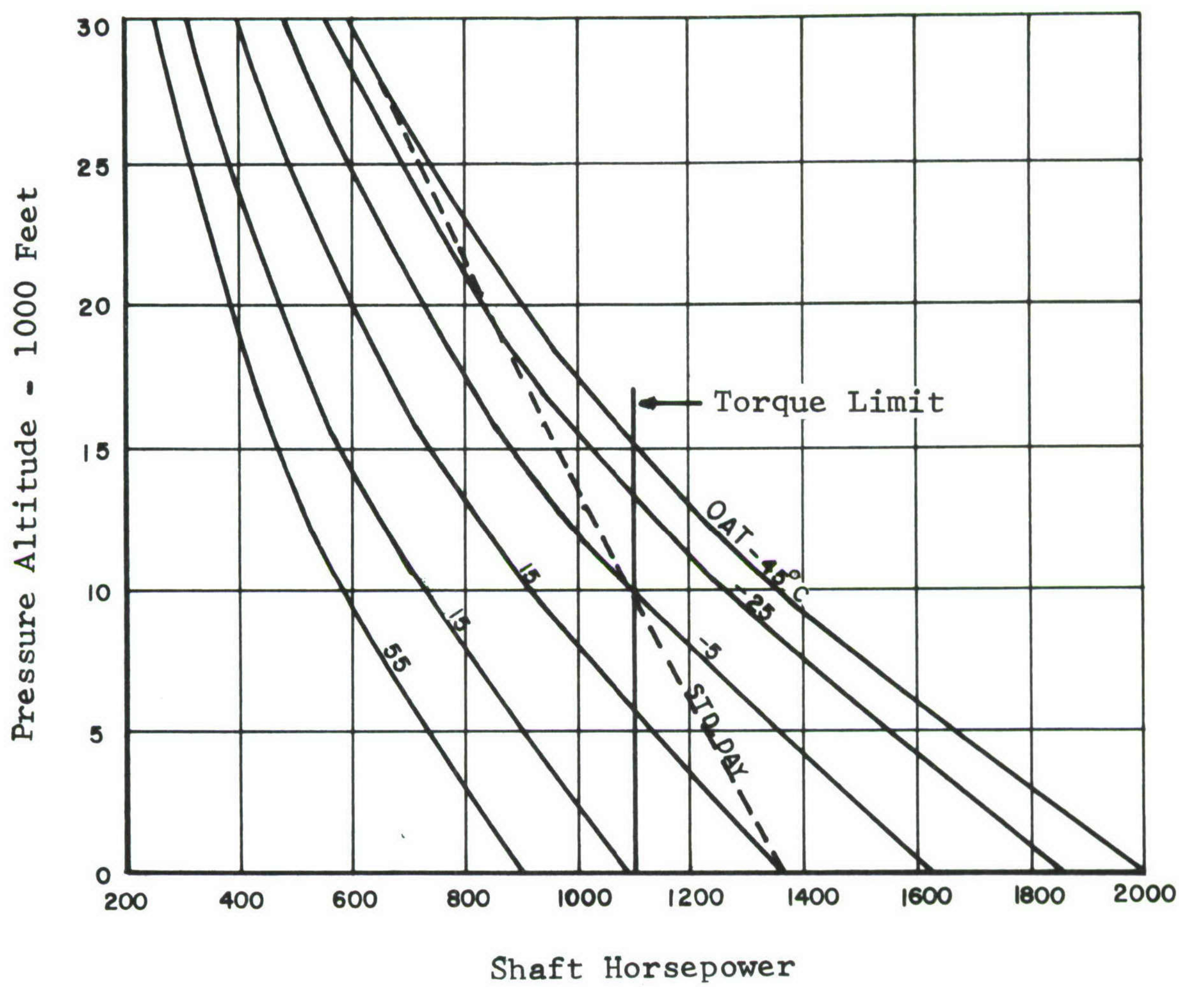

Figure 8. T53-L-13 Engine Operating Limits (Military Power).

computational procedure was utilized to obtain the absolute value of N_1, for any ambient conditions.

(1) Using Figure 8, obtain the SHP for the required ambient temperature and pressure condition.

(2) Compute the ratio of SHP/ $\delta\sqrt{\theta}$

(3) With the value of SHP/$\delta\sqrt{\theta}$ enter Figure 7 and obtain the corresponding value of $N_1/\sqrt{\theta}$

(4) Knowing $\sqrt{\theta}$ compute the required value of N_1

The resulting value of N_1 obtained in step (4) is that corresponding to the SHP at the required ambient condition of temperature and pressure specified in step (1).

The next step is to establish the power requirements for each flight mode considered. This is accomplished as follows:

(1) Knowing the relationships between SHP, temperature (θ) and pressure (δ), compute the maximum available OGE power coefficient (C_P), thus:

$$C_P = \frac{(SHP)550}{\rho A\ (\Omega R)^3}$$

Typical computations are shown in Table II for 5,000 ft. altitude.

(2) Using the value of C_P from step (1) above, enter the nondimensional hovering performance curves of Figure 9 and obtain the corresponding value of C_T, i.e. gross weight.

(3) Keeping C_T constant, enter Figure 9 and obtain the corresponding value of C_P for IGE hover at 2.0 ft. skid height.

100% N_1 = 25400 rpm Ω = 324 rpm = 33.9 rad/sec R = 24 ft
$A = \pi R^2 = 1809.56\ ft^2$ $C_p = 550\ SHP/\rho A(\Omega R)^3 \sigma$

$C_p = 0.563223 \times 10^{-12} \times SHP/\sigma$
Pressure Height = 5000 ft

1 MAXSHP	740	850	962	1070	1130	1200	1300
2 T^oC	55	40	30	20	15	10	0
3 T^oR	590.4	563.4	545.4	527.4	518.4	509.4	491.4
4 density alt.	10500	8900	7900	6750	6200	5700	4500
5 $\sigma \times 10^3$	1.729	1.810	1.875	1.942	1.976	2.020	2.080
6 δ	.832						
7 $\sqrt{\theta} = \sqrt{(3)/518.4}$	1.06723	1.0425	1.02571	1.0086	1.0	.983	.9736
8 $1/\delta\sqrt{\theta}$	1.1262	1.1510	1.17180	1.19167	1.2019	1.2227	1.2345
9 $C_p(OGE) \times 10^5$	24.1	26.4	28.89	31.33	32.20	33.46	35.2
10 $C_T \times 10^4$	31.7	33.9	36.2	38.4	39.2	40.4	41.8
11 $C_p(IGE) \times 10^5$ (2 ft hover)	19.7	21.4	23.2	25.0	25.6	26.7	28.0
12 SHP(IGE) 11x1/9	604	688	772	855	898	958	1036
13 $SHP(OGE)/\delta\sqrt{\theta}$ 1 x 8	833	979	1127	1275	1358	1467	1605
14 $SHP(IGE)/\delta\sqrt{\theta}$ 12 x 8	680	792	905	1019	1079	1171	1280
15 $N_1/\sqrt{\theta}$* (OGE)	23030	23625	24240	24850	25200	25630	26100
16 $N_1/\sqrt{\theta}$ (IGE)	22390	22875	23330	23800	24050	24425	24870
N_1(OGE)	24579	24650	24863	25064	25200	25194	25400**
% MAXN_1 (OGE)	96.77	97.0	97.89	98.68	99.21	99.19	100.0
N_1(IGE)	23895	23900	23930	24005	24050	24010	24200
% MAXN_1 (IGE)	94.07	94.1	94.21	94.50	94.68	94.52	95.3
ΔN_1 = N_1(OGE)-N_1(IGE)	684	750	933	1059	1150	1184	1200
%ΔN_1 reserve	2.69	2.95	3.67	4.17	4.53	4.66	4.7

** Based on extrapolation of referred data.

* Average of data for altitudes 0 15000 ft

Table II Typical Calculation of N_1

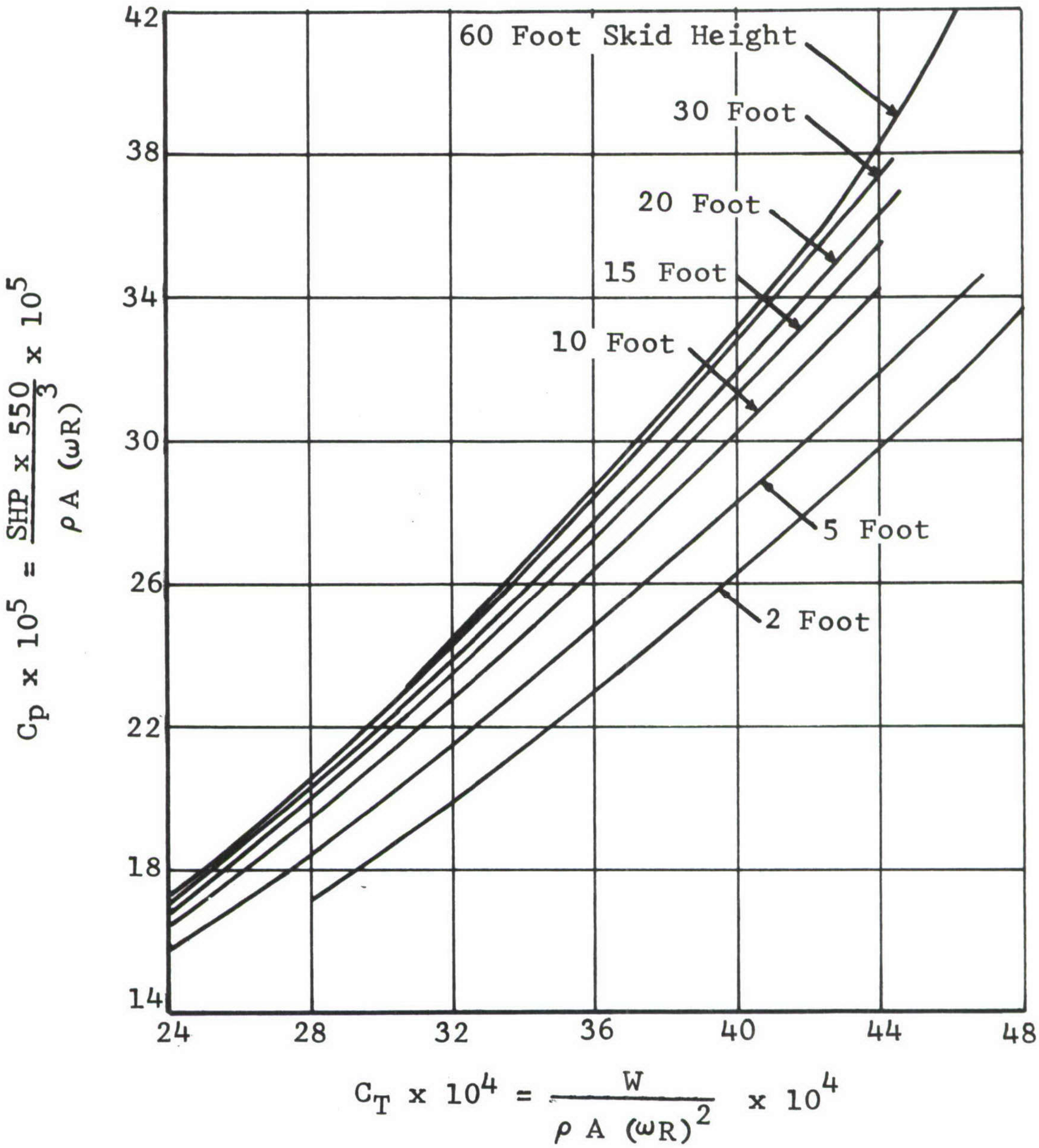

Figure 9. Non-Dimensional Hovering Performance Summary

Thus, both in and out of ground effect, power coefficients are established for a constant (maximum) value of C_T. Knowing the atmospheric conditions of temperature and pressure, air density is then computed and the corresponding values of SHP are determined.

The SHP for both the OGE and IGE conditions must be corrected for temperature and altitude and applied to the referred data curve of Figure 7. Corresponding values of referred N_1 are obtained and transformed into absolute values as shown for the sample case in Table II.

The final task is to determine the power requirement at the maximum gross weight at some forward velocity. Since it is somewhat difficult to fly the helicopter at a fixed velocity, it is desirable to find the range in which the power coefficient, C_P, remains essentially constant. Calculations show that this occurs in the range of 55 to 65 knots as indicated by Figure 10. This figure presents the referred N_1 versus forward speed (or values of tip speed ratio μ) for constant values of temperature. The results presented in Figure 10 were derived using the forward velocity performance data of Figure 11 and 12 applicable to the UH-1H helicopter. The computations presented in Table III are based on the values of C_T defined by C_P (OGE) and the appropriate values of advancing blade tip Mach number.

Thus, having established the power requirements for the

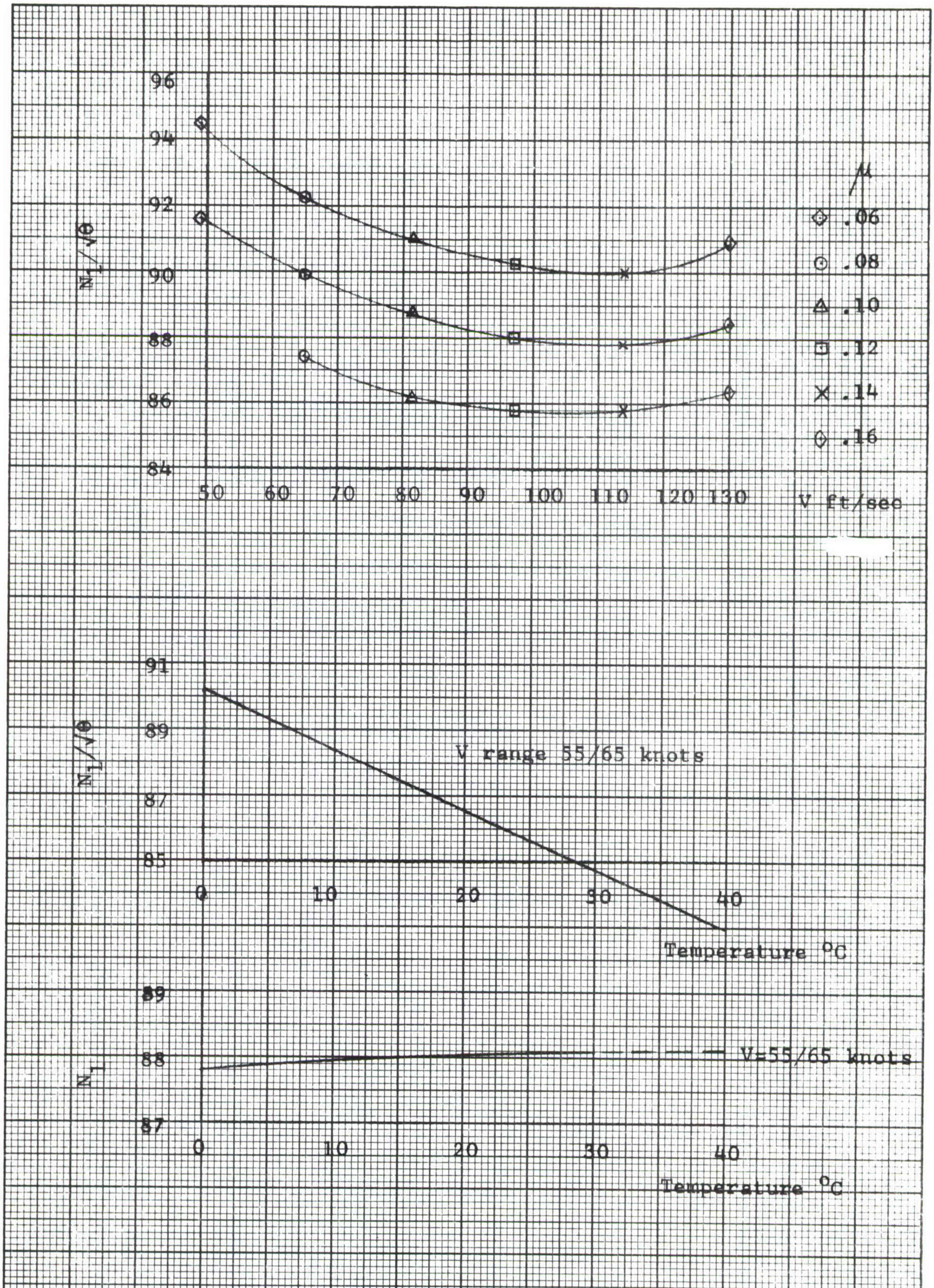

Figure 10. Referred N_1 Versus Velocity

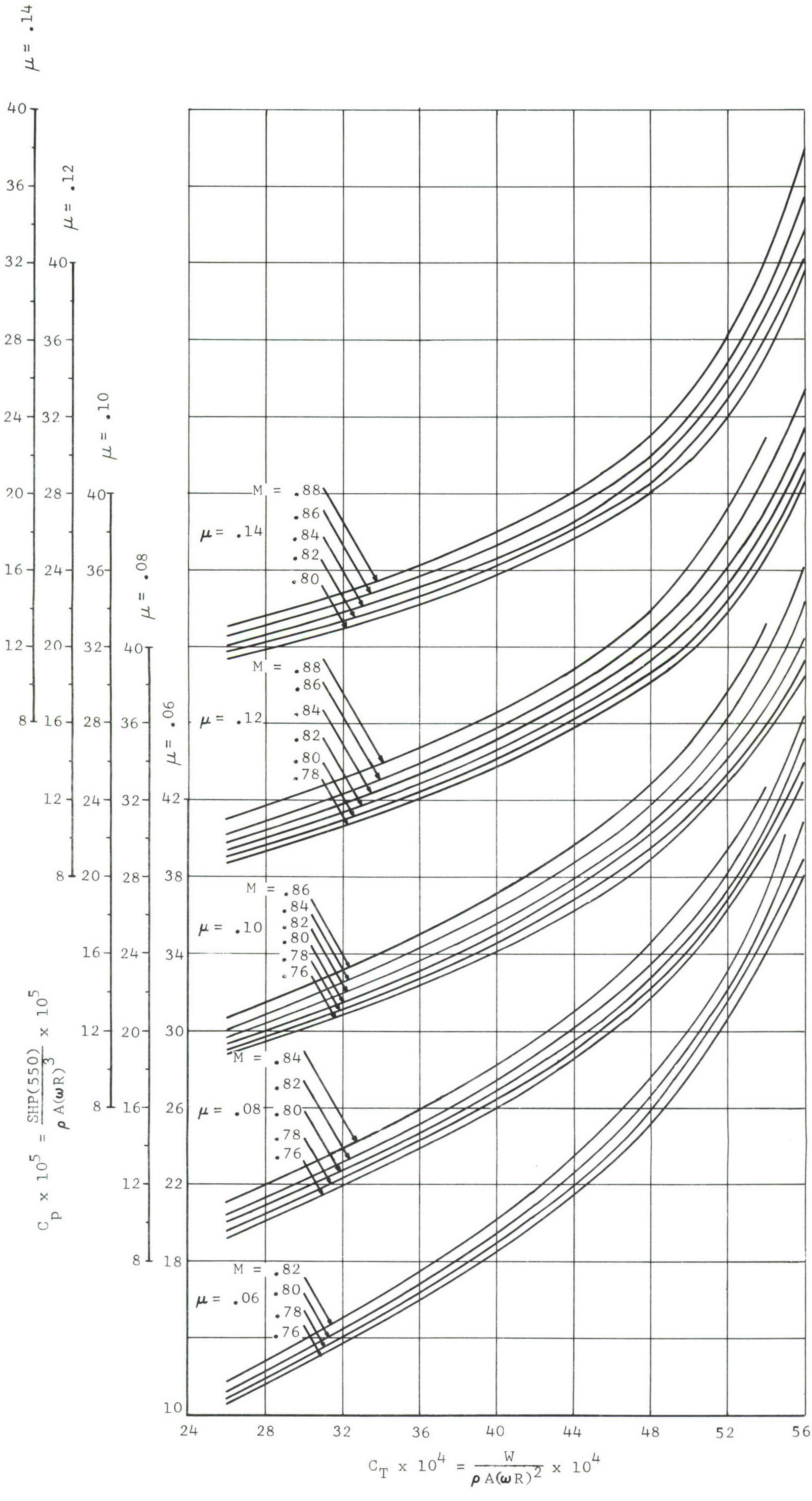

Figure 11. Nondimensional Level Flight Performance YUH-1H S/N 60-6029

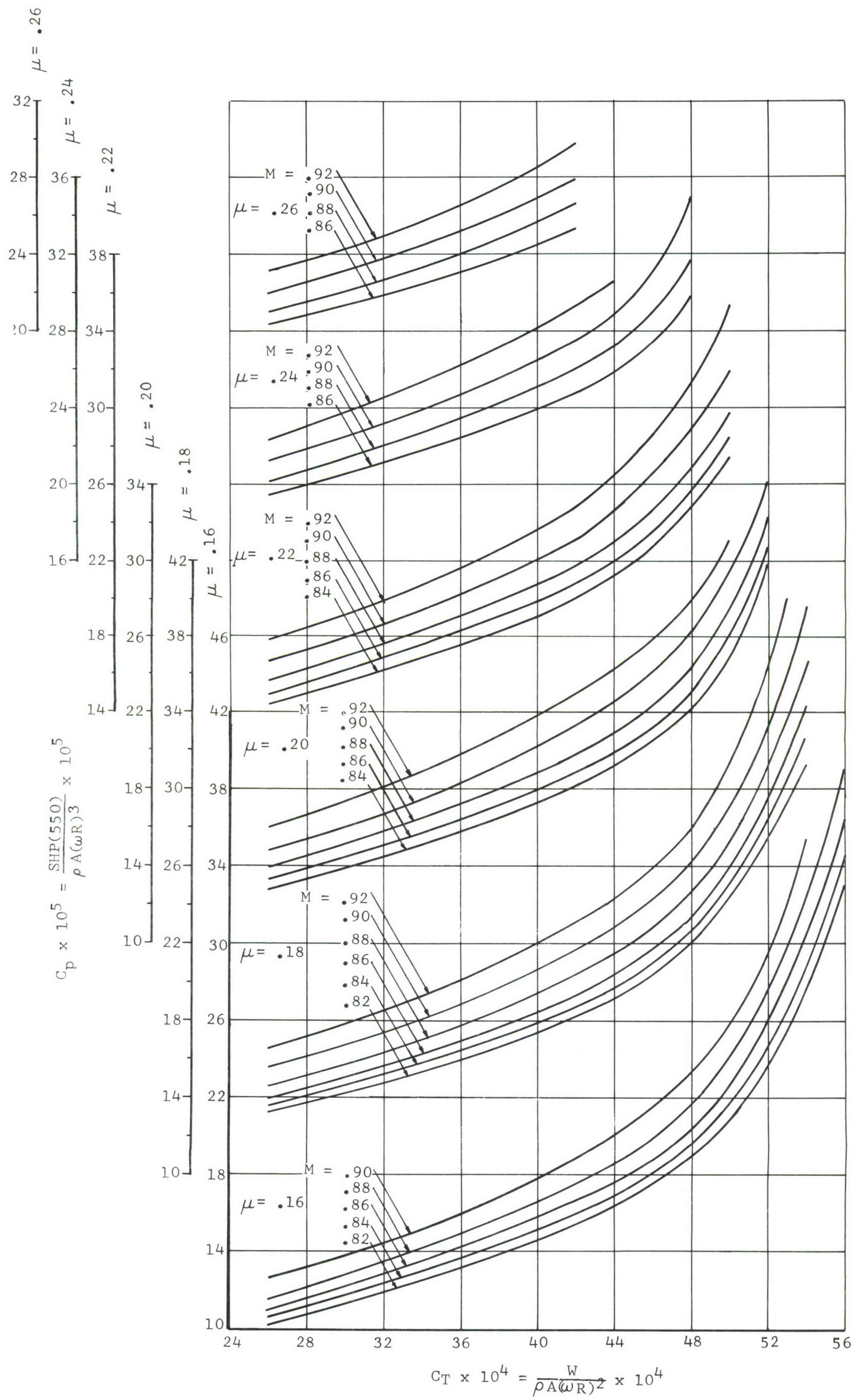

Figure 12. Nondimensional Level Flight Performance YUH-1H S/N 60-6029

<u>5000 ft</u> $V_f = M_T V_C - \Omega R$, $\Omega = 324$ rpm = 33.93 rad/sec, R = 24 ft

T = 30°C, $1/\delta\sqrt{\theta} = 1.1718$, $C_T = 36.2\times10^{-4}$, $V_c = 1145.33$

	μ	.06	.08	.10	.12	.14	.16
	M_T	-----	.768	.782	.796	.810	.824
	V_f ft/sec	-----	65.6	81.6	97.7	113.7	130
$C_p\times10^5$ @	μ and V_f	-----	16.3	14.7	14.1	14.1	14.85
	$SHP/\delta\sqrt{\theta}$	-----	634	572	549	549	518
	$N_1/\sqrt{\theta}$	-----	22220	21930	21820	21820	21950
	N_1	-----	22780	22493	22330	22381	22514
	$\%N_{1\ MAX}$	-----	89.6	88.6	88.1	88.1	88.6

T = 15°C, $1/\delta\sqrt{\theta} = 1.202$, $C_T = 39.2\times10^{-4}$, $V_c = 1116.6$

	M_T	.773	.787	.802	.816	.831	.845
	V_f	49.1	64.8	81.5	97.2	113.9	129.5
	C_p	21.1	18.6	16.4	16.0	15.9	16.65
	SHP/ θ	887	782	711	673	668	700
	N_1/ θ	23250	22825	22525	22375	22330	22475
	N_1	23250	22825	22525	22375	22330	22475
	$\%N_{1MAX}$	91.53	89.8	88.6	88.1	87.9	88.41

T = 0°C, $1/\delta\sqrt{\theta} = 1.2345$, $C_T = 41.9\times10^{-4}$, $V_e = 1087.16$

	M_T	.793	.808	.823	.838	.853	.868
	V_f	48.1	64.4	80.7	97.0	113.3	129.6
	C_p	24.2	21.0	19.2	18.2	18.1	19.2
	$SHP/\delta\sqrt{\theta}$	1071	930	850	806	801	850
	$N_1/\sqrt{\theta}$	24020	22343	23100	22920	22900	23100
	N_1	23386	22811	22490	22315	22295	22490
	$\%N_{1MAX}$	92.0	89.8	88.0	88.1	87.9	88.4

Table III. Typical Calculation of N_1

three flight modes at prevailing conditions corresponding to the existing gross weights, it is now possible to determine the power increments in terms of N_1 and the minimum reserve power, between the following flight modes:

(a) Hover out of ground effect (OGE)

(b) Hover in ground effect (IGE) at 2 ft. skid height

(c) Forward speed out of ground effect at V = 55-65 knots

In computing the values of N_1 for the above flight modes it is found that the N_1 versus temperature relationship is essentially independent of altitude, therefore the data for 5000 feet altitude can be used as representative for all conditions considered in this study. Thus, it is possible to relate the maximum N_1 for a particular engine at a given temperature to the required N_1 for the three flight modes.

Also, the data known as the N_1 bias curve was developed to reflect the engine over-temperature limitation. This limitation is implemented by a cam in the engine fuel system which trims the engine speed at ambient temperatures above 15°C in order to protect the engine from over-temperature.

The bias data is presented in Figure 13, which shows the change in the maximum N_1 from standard day as a function of temperature. By adding the calculated N_1's and the N_1 bias at given temperatures and subtracting the results from N_1 Standard Day (see Table IV), the required N_1 power curves for each condition are obtained. The final results are sum-

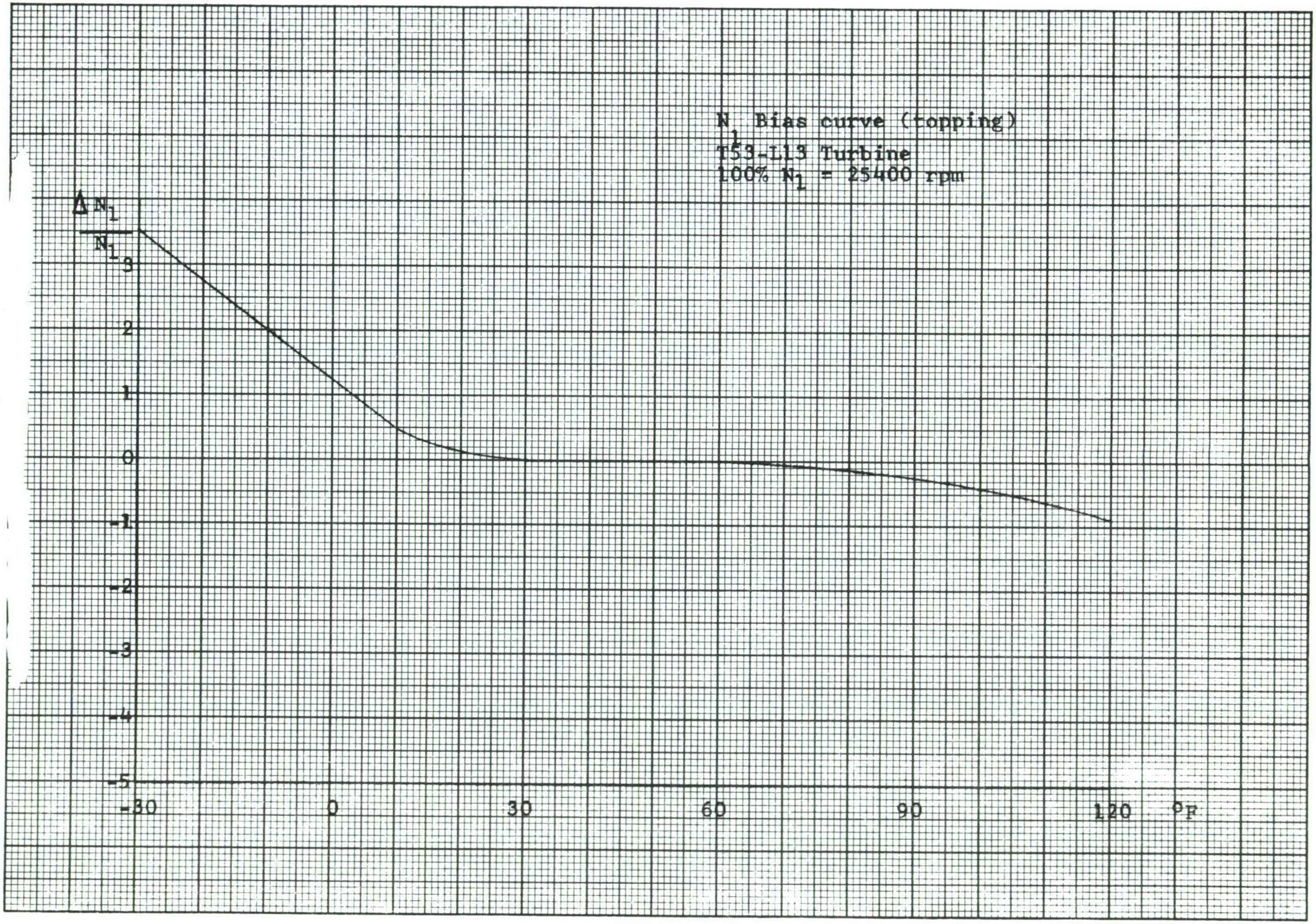

Figure 13. N_1 Bias Curve T-53-L13 Engine

T^oC	N_1(OGE)	N_1(IGE)	N_1(55/65KT)	N_1BIAS		
55	96.77	94.07	88.20	1.5		
40	97.0	94.1	88.10	.5		
30	97.89	94.21	88.05	.2		
15	99.21	94.68	88.00	0		
0	100.00	95.3	87.80	0		
	N_1 RESERVE (ΔN_1)					
	① OGE-IGE	② OGE-55/65	③ IGE-55/65	①+BIAS	②+BIAS	③+BIAS
55	2.70	8.57	5.87	4.20	10.07	7.37
40	2.90	8.90	6.00	3.40	9.4	6.50
30	3.68	9.84	6.16	3.88	10.04	6.36
15	4.53	11.21	6.68	4.53	11.21	6.68
0	4.70	12.20	7.5	4.70	12.20	7.50
	100% N_1 - (N_1 Reserve + Bias)					
55	95.80	89.93	92.63			
40	96.60	90.60	93.50			
30	96.12	89.96	93.64			
15	95.47	88.79	93.32			
0	95.30	87.80	92.50			

Data for 5000 ft is taken to be representative of the altitude range S.L. to 15000 ft

Table IV Maximum Usable N_1

marized in Figure 14.

For presentation purposes 100% is assumed as standard day N_1. In order to apply these curves to any UH-1H helicopter it is only necessary to align the standard day placard N_1 with the "Red Line" N_1 curve by shifting the ordinate scale upward. This will automatically bring the remaining curves to their correct value on the scale.

C. OPERATING INSTRUCTIONS

A summary of the operating instructions for the payload meter shown in Figure 6 is presented below.

Prior to operation the device must be zeroed for the specific helicopter in which it is installed.

1. Zero Adjust

Using Figure 15 the procedure for adjusting zero is as follows:

(a) Remove cover.

(b) Loosen the four (4) clamping screws on the N_1 scale plate.

(c) Set 15°C on temperature scale.

(d) Position N_1 scale plate such that the right edge of the red line (adjacent to N marker) is aligned with standard day N_1 from historical records.

(e) Tighten clamping screws.

(f) Replace cover.

2. Take Off

The following procedures are utilized to determine if

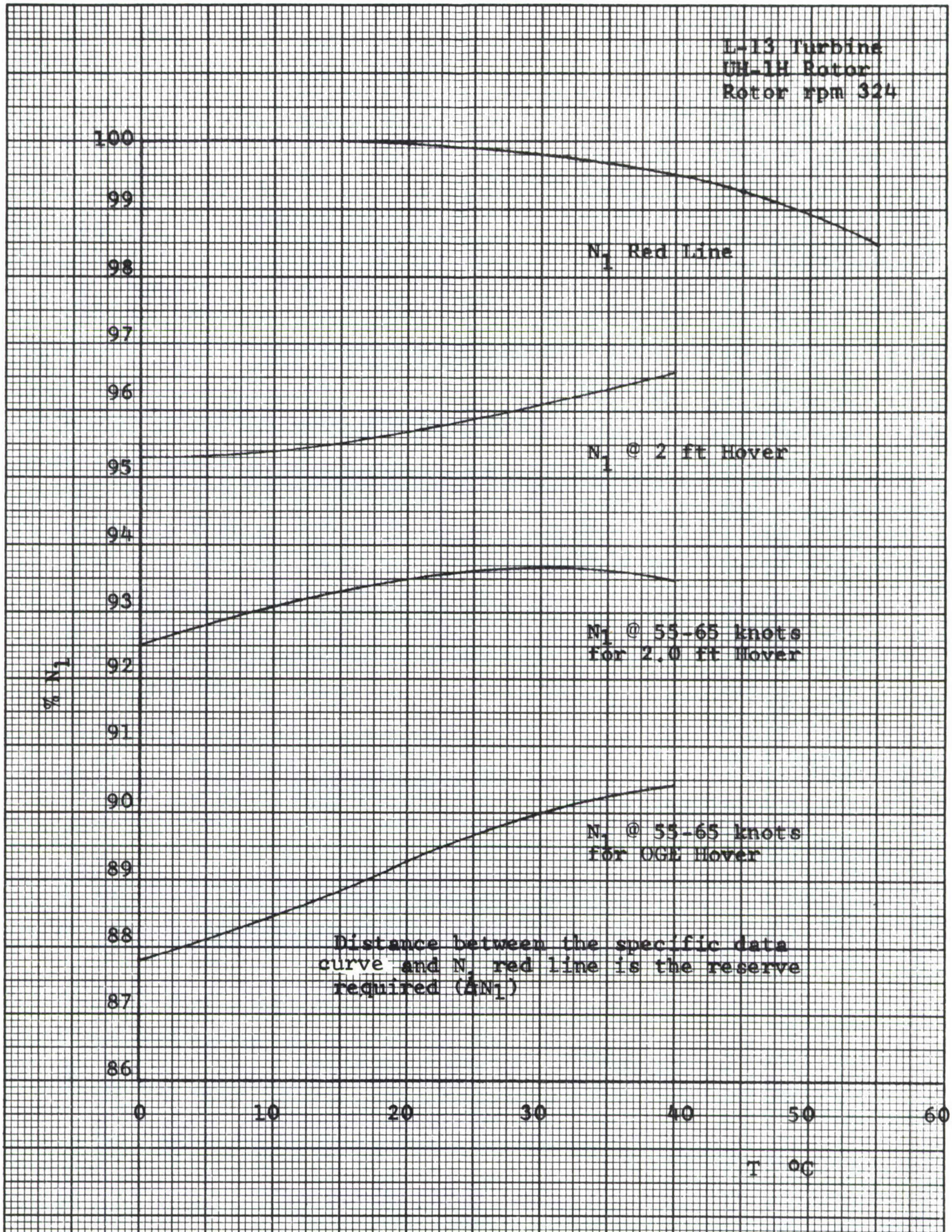

Figure 14. Maximum Usable N_1 Versus Temperature

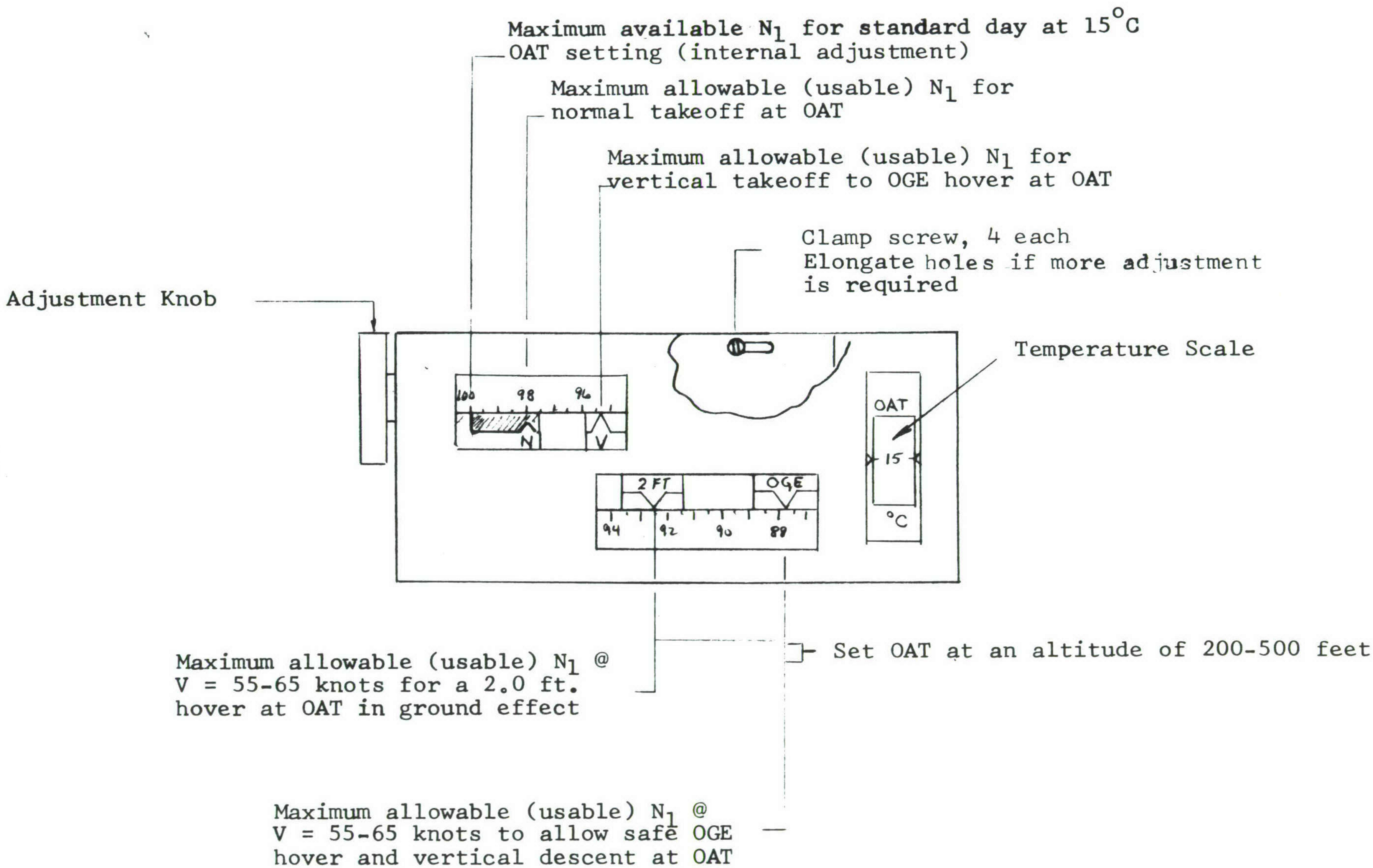

Figure 15. Go - No - Go Indicator - UH-1H Helicopter

sufficient power is available to safely execute a take-off.

(a) Check OAT and set indicator to read same.

(b) Check percent N_1 required to maintain a stabilized 2.0 ft. skid height hover.

(c) Relate the N_1 read in ((b) above) to that shown on the go-no-go indicator.

(d) Vertical take off - if the percent N_1 required to hover at 2.0 ft. does not exceed that shown on the indicator as (V) at the OAT there is sufficient power for a vertical take off.

(e) Normal take-off - if the percent N_1 required to hover at 2.0 feet does not exceed that shown on the indicator as (N) at the OAT there is sufficient power for a normal take off.

(f) Standard take-off techniques apply to intermediate power margins.

3. <u>Landing/Descent</u>

To determine if sufficient power is available to safely execute a landing the following procedures are utilized:

(a) Trim the aircraft to a level flight in the speed range between 55 to 65 knots.

(b) Check OAT and set indicator to read same.

(c) Relate the turbine N_1 to that shown on the indicator for the OAT.

(d) Vertical descent and landing:

(i) If the turbine N_1 does not exceed that shown on the indicator as OGE at the OAT the aircraft has sufficient power for a vertical descent and landing.

(ii) If the turbine N_1 is greater than that shown as OGE but less than that shown as (2 ft), the aircraft has sufficient power for a 2.0 ft. skid height hover. However, a descent other than vertical is required.

NOTE 1: The landing descent maneuvers should be performed at an altitude of approximately 200-500 ft. to ensure minimum deviation of OAT from that indicated by the meter.

NOTE 2: The payload meter described herein does not in any way affect the operation of the aircraft. Therefore, the warning and caution notes applicable to the operation of the aircraft are not affected.

D. FLIGHT TEST EVALUATION OF THE GO-NO-GO PAYLOAD INDICATOR

The GO-NO-GO payload indicator as described above was flight tested by the U. S. Army Aviation Test Board (USAAVNTBD) to determine whether the instrument had military potential. The tests were conducted at a variety of air temperature and density altitudes at Fort Rucker, Alabama and Asheville, North Carolina.

The data obtained from the instrument and from the air-

craft engine instruments during the flight tests were compared. The GO-NO-GO payload indicator provided the pilot with a realistic comparison of power required with power available from which the capability for take-offs and landings could be predicted. The accuracy of the instrument, although found to be somewhat conservative, was acceptable throughout the flight conditions investigated.

It was therefore concluded that the GO-NO-GO payload indicator has a military potential and recommendations were made for further development of this instrument.

IV. CONCLUSIONS AND RECOMMENDATIONS

Based on the results of the study presented in this report, the following conclusions and recommendations are made:

1. The results of the feasibility study indicate that an automatic helicopter payload capability meter is feasible.
2. The accuracy of such an instrument is primarily a function of the accuracy of the onboard instrumentation to obtain ambient air temperature and pressure altitude inputs. Based on the analysis of the performance data applicable to the CH-47A helicopter it is estimated that with the presently available instrumentation a payload measurement within $\pm$ 251 lb is possible.
3. The accuracy of the instrument can be considerably improved by improving the accuracy of the helicopter instrumentation and by utilizing a digital instead of analog approach in calculating helicopter payload. Utilizing this approach, a payload measurement accuracy within $\pm$ 200 lb is achievable for the CH-47A helicopter.
4. In-flight measurement of aircraft initial weight is also feasible. This could be achieved by performing a simple hovering test (OGE).
5. The GO-NO-GO manual payload meter indicator designed for the UH-1H helicopter provided the pilot with valuable payload information from which the aircraft capability for take offs and landings could be predicted under a variety of atmospheric conditions.

6. Based on the flight test data obtained by the U. S. Army Aviation Test Board it was concluded that the GO-NO-GO payload meter has a military potential.
7. Based on the results of this study it is recommended that further development work be performed to perfect the GO-NO-GO payload indicator.
8. It is further recommended that an automatic payload capability meter be developed, installed, and flight tested on a helicopter, to provide the pilot with an instantaneous and automatic display of helicopter lifting capability under a variety of atmospheric conditions.

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U. S. Army Land Warfare Laboratory
Aberdeen Proving Ground
Maryland, 21005

Attention: CRDLWL-6C
Mr. R. P. McGowan

Subject: Contract DAAD05-68-C-0366
Work Assignment No. 2 (Modified)

Reference: (a) LWL letter CRDLWL-6C dated 28 January 1971

Enclosure: (1) Final Reproducible Report (DCR-326)

Gentlemen:

Enclosure (1), incorporating the recommended changes set forth under reference (a), is forwarded herewith.

Very sincerely yours,

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13. ABSTRACT

A feasibility study was made of a helicopter payload meter concept. A simple, manually operated device was developed and tested, which gives an indication of payload capability in terms of gas generator speed for the prevailing atmospheric conditions where vertical take-offs and landings are required from a confined area. Tests were conducted by the U.S. Army Aviation Test Board, and the device was found to have "military potential."

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